THE NINEVAH MISSION: ANALYTICAL ANALYSES AND CALCULATIONS FOR TRAJECTORY AND PROPULSION
Volume 2

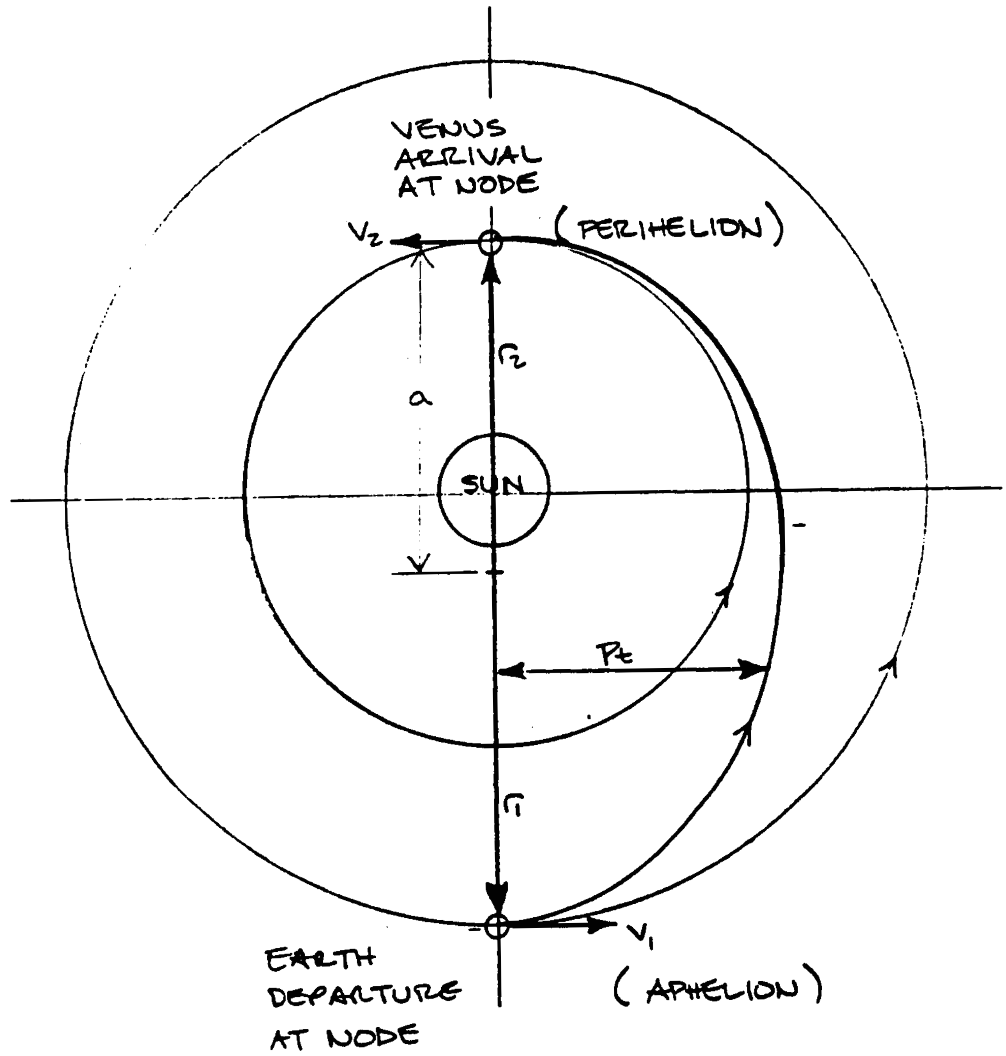
A design project by students in the Department of Aerospace Engineering at Auburn University, Auburn, Alabama, under the sponsorship of NASA/USRA Advanced Design Program.

Auburn University
Auburn, Alabama
June, 1988

TABLE OF CONTENTS

TRAJECTORY ANALYSIS

THE HELIOCENTRIC TRAJECTORY OF NINEVAH IS SHOWN BELOW. THE HOHMANN TRANSFER CARRIES THE SPACECRAFT FROM EARTH AT APHELION TO VENUS AT PERIHELION. PLANETARY ORBITS ARE ASSUMED CIRCULAR.



TRANSFER BEGINS AT ONE NODE (POINT WHERE ECLIPTIC PLANE INTERCEPTS VENUS' ORBITAL PLANE) AND ENDS AT THE OTHER NODE 180° LATER. THE SUN LIES ON THE LINE OF NODES WHICH IS THE LINE

OF INTERSECTION OF THE TWO HELIOCENTRIC ORBITAL PLANES. THE FOLLOWING ARE HELIOCENTRIC DATA OF THE HOHMANN ELLIPSE, MOST OF WHICH IS USED TO CALCULATE THE HELIOCENTRIC SPEED OF EARTH, VENUS, AND PROBE; THESE SPEEDS DETERMINE THE ΔV'S FOR HELIOCENTRIC TRANSFER.

- SEMI-MAJOR AXIS (a_t):

$$2a_t = r_1 + r_2 \qquad \text{(BATE, ET AL: p. 30)} \qquad (1)$$
$$= (149.5 + 108.1) \times 10^6 \text{ km}$$
$$a_t = 128.8 \times 10^6 \text{ km}$$

- ECCENTRICITY (e_t):

$$e_t = \frac{r_1 - r_2}{r_1 + r_2} \qquad \text{(BATE, ET AL: p. 31)} \qquad (2)$$
$$= \frac{(149.5 - 108.1)}{(149.5 + 108.1)}$$
$$e_t = 0.1607$$

- SEMI-LATUS RECTUM (p_t):

$$p_t = a_t(1 - e_t^2) \qquad \text{(BATE, ET AL: p. 24)} \qquad (3)$$
$$= (128.8 \times 10^6 \text{ km})[1 - (0.1607)^2]$$
$$p_t = 1.254732 \times 10^8 \text{ km}$$

- SPECIFIC MECHANICAL ENERGY ($\mathcal{E}_t$):

$$\mathcal{E}_t = -\frac{\mu_\odot}{2a_t} \qquad \text{(BATE, ET AL: p. 28)} \qquad (4)$$
$$= -\frac{1.3271544 \times 10^{11} \text{ km}^{\cancel{3}\,2}/\text{sec}^2}{2(128.8 \times 10^6 \cancel{\text{km}})}$$
$$\mathcal{E}_t = -515.1997 \text{ km}^2/\text{sec}^2$$

THE TERM $\mu_\odot$ IS THE GRAVITATIONAL PARAMETER OF THE SUN AND APPEARS ALSO IN THE SPECIFIC ANGULAR MOMENTUM AS SHOWN BELOW.

- SPECIFIC ANGULAR MOMENTUM (h_t):

$$h_t = \sqrt{p_t \mu_\odot} \qquad \text{(BATE, ET AL: p.167)} \qquad (5)$$

$$= \sqrt{(1.254732 \times 10^8\ km)(1.3271544 \times 10^{11}\ km^3/sec^2)}$$

$$h_t = 4.0807221 \times 10^9\ \frac{km^2}{sec}$$

THE TIME OF TRANSFER OR TIME OF FLIGHT (TOF) FOR THE TRIP TO VENUS IS HALF THE PERIOD OF AN ELLIPSE,

$$TOF = \pi \sqrt{\frac{a_t^3}{8\mu_\odot}} \qquad \text{(BATE, ET AL: p.165)} \qquad (6)$$

$$= \pi \sqrt{\frac{(128.8 \times 10^6\ km)^3}{8(1.3271544 \times 10^{11}\ km^3/sec^2}}$$

$$= 12{,}605{,}583.16\ sec$$

$$TOF = 145.89\ days \quad (4.8\ months).$$

THE HELIOCENTRIC SPEED OF THE PROBE UNDER EARTH'S SPHERE OF INFLUENCE (SOI) IS AS FOLLOWS:

$$v_1 = \sqrt{2\left(\frac{\mu_\odot}{r_1} + \mathcal{E}_t\right)} \qquad \text{(BATE, ET AL: p.164)} \qquad (7)$$

$$= \sqrt{2\left(\frac{1.3271544 \times 10^{11}\ km^3/sec^2}{149.5 \times 10^6\ \cancel{km}} - 515.1997\ \frac{km^2}{sec^2}\right)}$$

$$v_1 = 27.2958\ \frac{km}{sec}.$$

EARTH'S ORBITAL SPEED, V_{cs1}, IS SIMPLY A FUNCTION OF THE SUN'S GRAVITATIONAL PARAMETER AND EARTH'S RADIAL DISTANCE FROM THE SUN'S CENTER.

$$V_{cs1} = \sqrt{\frac{\mu_\odot}{r_1}} \qquad \text{(BATE, ET AL: p. 165)} \qquad (8)$$

$$= \sqrt{\frac{1.3271544 \times 10^{11}\ km^3/sec^2}{149.5 \times 10^6\ km}}$$

$$V_{cs1} = 29.7948\ \frac{km}{sec}$$

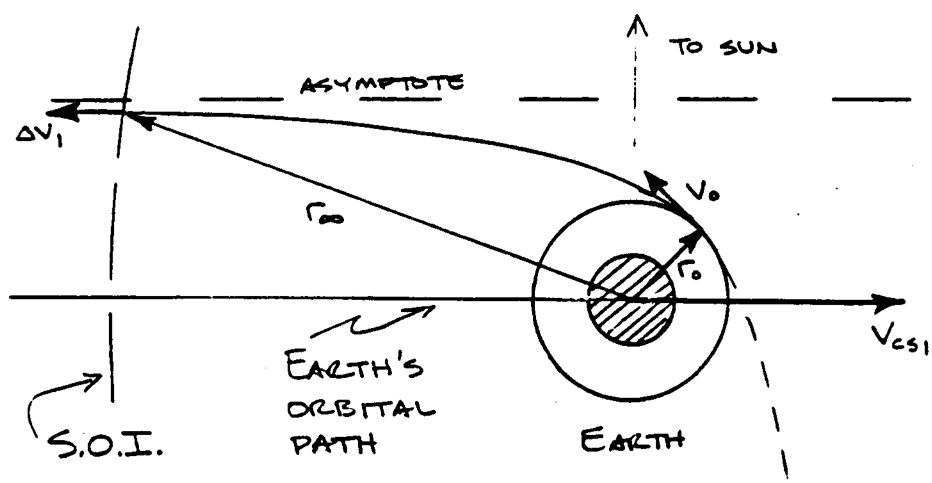
THE ΔV REQUIRED TO LEAVE EARTH ON A HELIOCENTRIC HOHMANN TRAJECTORY IS THE DIFFERENCE BETWEEN THE HELIOCENTRIC TRANSFER SPEED, V_1, AND EARTH'S ORBITAL SPEED, V_{cs1}. THIS DIFFERENCE OR ΔV_1 IS THE "ENERGY" REQUIRED JUST BEYOND EARTH'S SOI TO START THE HELIOCENTRIC JOURNEY.

$$\Delta V_1 = V_1 - V_{cs1} \qquad \text{(BATE, ET AL: p. 165)} \qquad (9)$$

$$= (27.2958 - 29.7948)\ \frac{km}{sec}$$

$$\Delta V_1 = (-)\ 2.499\ \frac{km}{sec}$$

THE NEGATIVE SIGN INDICATES THAT THE PROBE IS LAUNCHED IN THE DIRECTION OPPOSITE THE EARTH'S ORBITAL VELOCITY AS SHOWN BELOW.



NINEVAH'S EARTH DEPARTURE FOLLOWS A HYPERBOLIC PATH THAT ORIGINATES FROM PERIGEE WHERE r_0 IS 43,392.912 km. THE PROBE IS IN A CIRCULAR ORBIT UPON WHICH ITS SPEED IS CALCULATED USING EQUATION (8),

$$V_{cs_0} = \sqrt{\frac{\mu_\oplus}{r_0}}, \quad \mu_\oplus \equiv \text{EARTH'S GRAVITATIONAL PARAMETER} \tag{10}$$

$$= \sqrt{\frac{3.986012 \times 10^5\ km^3/sec^2}{43{,}392.912\ km}}$$

$$V_{cs_0} = 3.0308\ \frac{km}{sec}$$

THE REQUIRED INJECTION SPEED, V_0, TO OBTAIN ΔV_1 AT THE SOI IS FOUND AS FOLLOWS,

$$V_0 = \sqrt{V_\infty^2 + \frac{2\mu_\oplus}{r_0}} \quad \text{(BATE, ET AL: p. 369)}. \tag{11}$$

ASSUMING THAT $V_\infty = \Delta V_1$ AT SOI,

$$V_0 = \sqrt{\left(2.499\ \frac{km}{sec}\right)^2 + \frac{2\,(3.986012 \times 10^5\ km^3/sec^2)}{43{,}392.912\ km}}$$

$$= 4.9615\ \frac{km}{sec}.$$

THEN, THE ΔV REQUIRED AT PERIGEE TO GIVE THE NINEVAH SPACECRAFT THE REQUIRED "ΔV_1" AT SOI FOR HELIOCENTRIC TRANSFER IS SIMPLY,

$$\Delta V_0 = V_0 - V_{cs_0} \tag{12}$$

$$\boxed{\underline{\underline{\Delta V_0 = (+)\,1.9307\ \frac{km}{sec}}}}.$$

NOW OUTSIDE EARTH'S SOI AND ITS HELIOCENTRIC SPEED REDUCED BY 2.499 $\frac{km}{sec}$, NINEVAH FALLS TOWARD THE SUN ALONG THE

ELLIPTICAL TRAJECTORY ABOUT THE SUN UNTIL THE SPACECRAFT ENCOUNTERS THE CYTHEREAN SOI. FROM EQUATION (7), NINEVAH'S SPEED RELATIVE TO THE SUN IS,

$$V_2 = \sqrt{2\left(\frac{\mu_\odot}{r_2} + \mathcal{E}_t\right)} \tag{13}$$

$$= \sqrt{2\left(\frac{1.3271544 \times 10^{11}\ km^3/sec^2}{108.1 \times 10^6\ km} - 515.1997\ \frac{km^2}{sec^2}\right)}$$

$$V_2 = 37.7494\ km/sec$$

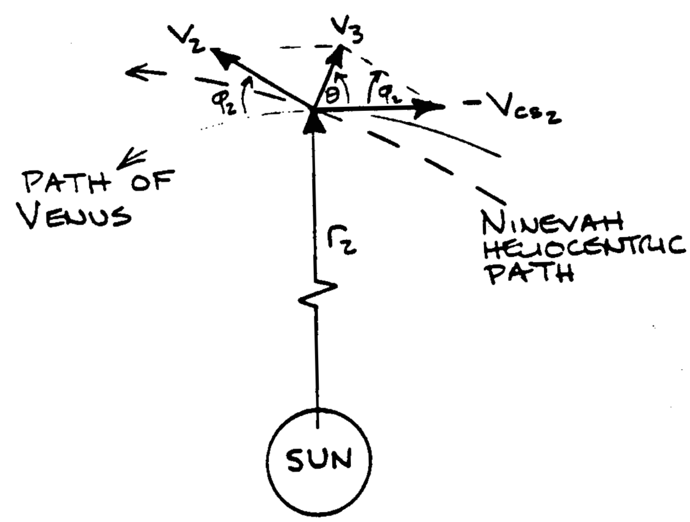
VENUS' ORBITAL SPEED IS OBTAINED FROM THE FORM OF EQUATION (8),

$$V_{cs_2} = \sqrt{\frac{\mu_\odot}{r_2}} \tag{14}$$

$$= \sqrt{\frac{1.3271544 \times 10^{11}\ km^3/sec^2}{108.1 \times 10^6\ km}}$$

$$V_{cs_2} = 35.0387\ \frac{km}{sec}$$

APPROACH TO VENUS ALONG THE TRANSFER PATH IS SHOWN BELOW.



ORIGINAL PAGE IS
OF POOR QUALITY

To verify that the transfer is tangent to Venus' orbital path, substituting into equation (15), which relates distance and velocity to the heliocentric specific angular momentum, gives the angle of intercept φ_2 as,

$$
\begin{aligned}
h_t &= r_2 V_2 \cos\varphi_2 \qquad \text{(Bate, et al: p. 373)} \qquad (15) \\
\cos\varphi_2 &= \frac{h_t}{r_2 V_2} \\
&= \frac{4.0807221 \times 10^9 \text{ km}^2/\text{sec}^2}{(108.1 \times 10^6 \text{ km})(37.7494 \text{ km/sec})} \\
&= 1.0000 \\
\varphi_2 &= 0^\circ .
\end{aligned}
$$

By the law of sines, the angle θ, which indicates Ninevah's angular relationship with respect to Venus, is found to be

$$
\begin{aligned}
\sin\theta &= \frac{V_7}{V_3} \sin\varphi_2 \qquad \text{(Bate, et al: p.374)} \qquad (16) \\
\theta &= 0^\circ .
\end{aligned}
$$

Obviously, the angles φ_2 and θ, both of which are zero, show the heliocentric transfer path to be tangent at Venus encounter.

Ninevah's speed relative to Venus is a simple application of the law of cosines,

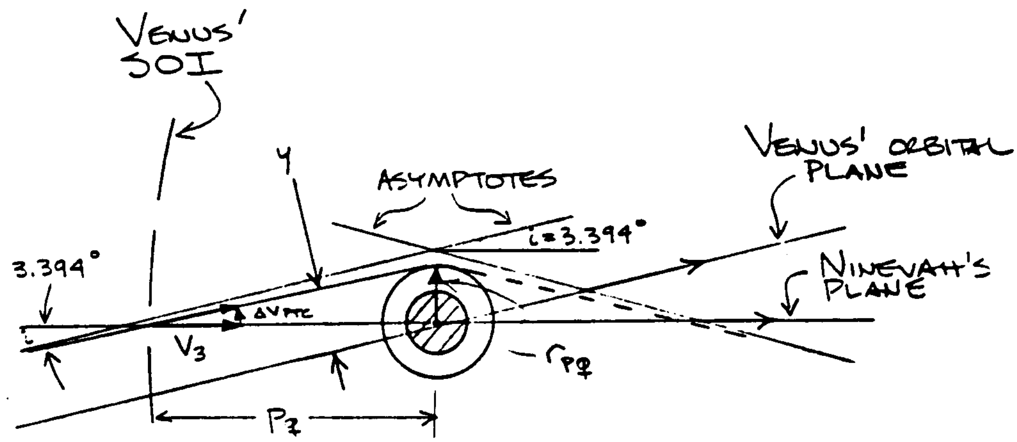
AS GIVEN IN EQUATION (17) BY BATE, ET AL (p. 373),

$$V_3^2 = V_2^2 + V_{cs_2}^2 - 2V_2V_{cs_2}\cos\varphi_2 \tag{17}$$

$$= (37.7494 \tfrac{km}{sec})^2 + (35.0387 \tfrac{km}{sec})^2 - 2(37.7494 \tfrac{km}{sec}) * (35.0387 \tfrac{km}{sec})(1)$$

$$V_3 = 2.7107 \text{ km/sec}.$$

THIS SPEED, V_3, IS NINEVAH'S SPEED UPON ENCOUNTER WITH THE CYTHEREAN SOI AS SHOWN BELOW.



THE PLANE CHANGE UPON ARRIVAL AT CYTHEREAN SOI IS SMALL DUE TO THE LOW INCLINATION OF THE CYTHEREAN ORBITAL PLANE TO THE TRANSFER PLANE. THE ENERGY REQUIREMENT TO MAKE THIS CHANGE IS FOUND FROM EQUATION (18),

$$\Delta V_{PTC} = 2V_3 \sin\frac{i}{2} \quad \text{(BATE, ET AL: p. 379)} \tag{18}$$

$$= 2(2.7107 \text{ km/sec}) \sin\left(\frac{3.394°}{2}\right)$$

$$\boxed{\Delta V_{PTC} = 0.16055 \text{ km/sec}}.$$

THE SEPARATION DISTANCE, y, WHICH IS THE DISTANCE (MEASURED AT SOI) BETWEEN THE CYTHEREAN ORBITAL PLANE AND NINEVAH'S PLANE OF TRANSIT, IS CALCULATED USING THE SIMPLE GEOMETRICAL EQUATION BELOW.

$$y = R_{s♀} \sin i \tag{19}$$

$R_{s♀}$ IS THE RADIUS OF CYTHEREAN SOI AND IS A FUNCTION OF THE MASSES OF VENUS AND THE SUN AND MEAN DISTANCE, D_m, SEPARATING THE BODIES. FROM BATE, ET AL (p. 333),

$$R_{s♀} = \left(\frac{m_♀}{M_☉}\right)^{2/5} D_m \tag{20}$$

$$= \left(\frac{0.817}{333432}\right)^{2/5} (108.1 \times 10^6 \text{ km})$$

$$R_{s♀} = 615{,}901.9876 \text{ km} .$$

THUS,

$$y = (615{,}901.9876 \text{ km}) \sin (3.394°)$$

$$y = 36{,}462.5296 \text{ km} .$$

THE SEPARATION DISTANCE IS USED TO CALCULATE THE ELEMENTS OF THE HYPERBOLIC APPROACH, WHICH ARE THEMSELVES USED TO DETERMINE THE RADIUS AND VELOCITY OF PERICYTH, $r_{p♀}$ AND $V_{p♀}$, RESPECTIVELY. AFTER $r_{p♀}$ AND $V_{p♀}$ ARE KNOWN, A TRADE STUDY ON TYPE OF ORBIT FOR VENUS OBSERVATION IS DONE TO DETERMINE THE BEST

ORBIT IN TERMS OF MINIMUM ENERGY REQUIREMENT (MER) AND MISSION OBJECTIVE.

FOR THE HYPERBOLIC APPROACH, THE SPECIFIC MECHANICAL ENERGY FROM EQUATION (21) IS,

$$\mathcal{E}_\varphi = \frac{V_3^2}{2} - \frac{\mu_\varphi}{r_\infty} \tag{21}$$

$$= \frac{(2.7107\ \mathrm{km/sec})^2}{2} - \cancel{\frac{\mu_\varphi}{r_\infty}}^{\,0}$$

$$\mathcal{E}_\varphi = 3.673947\ \frac{\mathrm{km}^2}{\mathrm{sec}^2}.$$

THE SECOND TERM IN EQUATION GOES TO ZERO AS r_∞ (OR $R_{s\varphi}$) IS INFINITELY LARGE COMPARED TO THE OTHER PARAMETERS IN THE EQUATION.

THE SPECIFIC ANGULAR MOMENTUM, h_φ, IS FOUND USING 'y' AS THE MOMENTUM ARM AS FOLLOWS,

$$h_\varphi = y V_3 \qquad \text{(BATE, ET AL: p. 376)} \tag{22}$$

$$= (36{,}462.5296\ \mathrm{km})(2.7107\ \mathrm{km/sec})$$

$$h_\varphi = 98{,}838.9425\ \mathrm{km^2/sec}.$$

THE SEMI-LATUS RECTUM IS FOUND FROM REARRANGING EQUATION (3) TO GET,

$$P_\varphi = \frac{h_\varphi^2}{\mu_\varphi} \tag{23}$$

$$= \frac{(98{,}838.9425\ \mathrm{km^2/sec})^2}{3.257 \times 10^5\ \mathrm{km^3/sec^2}}$$

$$P_\varphi = 29{,}994.2786\ \mathrm{km}.$$

THE ECCENTRICITY OF THE HYPERBOLIC APPROACH IS DETERMINED FROM EQUATION (24) AS FOLLOWS FROM BATE, ET AL: p. 376,

$$e_{♀} = \sqrt{1 + 2\mathcal{E}_{♀} h_{♀}^2 / \mu_{♀}^2} \qquad (24)$$

$$= \sqrt{1 + 2(3.673947)(29{,}994.2786)/(3.257 \times 10^5)}$$

$$e_{♀} = 1.2949 \,.$$

FINALLY, THE RADIUS AND VELOCITY AT PERICYTH FROM THE HYPERBOLIC TRAJECTORY ARE AS FOLLOWS:

$$r_{p♀} = \frac{p_{♀}}{1+e_{♀}} \qquad \text{(BATE, ET AL: p. 376)} \qquad (25)$$

$$= \frac{29{,}994.2786 \text{ km}}{1 + 1.2949}$$

$$\underline{\underline{r_{p♀} = 13{,}070.1614 \text{ km}}} \;;$$

AND REARRANGING EQUATION (15) GIVES

$$V_{p♀} = \frac{h_{♀}}{r_{p♀}} \qquad (26)$$

$$= \frac{98{,}838.9425 \text{ km}^2/\text{sec}}{13{,}070.1614 \text{ km}}$$

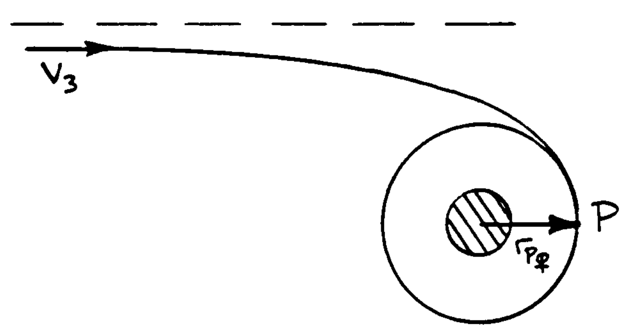
$$\underline{\underline{V_{p♀} = 7.5622 \frac{\text{km}}{\text{sec}}}} \,.$$

NOW CONSIDER THE BEST TYPE OF ORBITAL TRANSFER TO OBTAIN CYTHEREAN ORBIT AND THE BEST CYTHEREAN ORBIT, BASING BOTH ON MER AND PRIMARY MISSION OBJECTIVE (MAPPING).

ORBITAL TRADE STUDY

THIS SECTION COMPARES CIRCULAR ORBIT ACQUISITION WITH ELLIPTICAL ORBIT ACQUISITION; AND CONCLUSIONS ARE DRAWN REGARDING THE BEST BALANCE BETWEEN MER (LOW ΔV'S) AND THE MISSION OBJECTIVE INVOLVING SURFACE MAPPING.

(1) CIRCULAR ORBIT IS OBTAINED BY A TOTAL OF 3 BURNS, THE SEQUENCE IS AS FOLLOWS:



$r_{P♀} = 13{,}070.1614$ km

$V_{P♀}' = 7.5622$ km/sec

FROM EQN. (10), $V_{cs} = \sqrt{\frac{\mu_♀}{r_{P♀}}}$ (27)

$$= \sqrt{\frac{3.257 \times 10^5 \text{ km}^3/\text{sec}^2}{13{,}070.1614 \text{ km}}}$$

$$V_{cs} = 4.9919 \text{ km/sec}$$

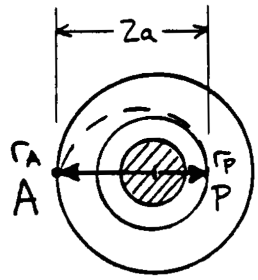
SO THAT $\Delta V_{P♀} = V_{cs} - V_{P♀}$ (28)

$$= (4.9919 - 7.5622) \text{ km/sec}$$

$$\underline{\underline{\Delta V_{P♀} = (-)\,2.5703 \text{ km/sec}}}.$$

ΔV_P IS APPLIED RETROACTIVELY TO SLOW CRAFT INTO AN INTERMEDIATE CIRCULAR ORBIT OF

13,070.1614 km. NOW TO PLACE CRAFT INTO INSTRUMENT MAPPING RANGE, USE HOHMANN TRANSFER:



$$r_A = 13{,}070.1614 \text{ km}$$

$$r_P = 6587 \text{ km}$$
(400 km DESIRED ALT.)

$$2a = r_a + r_p = 9828.5807 \text{ km}$$

$$V_{cs_A} = 4.9919 \text{ km/sec}$$

$$V_{cs_P} = 7.0318 \text{ km/sec}$$

FROM EQN. (4),

$$\mathcal{E} = -\frac{\mu_\oplus}{2a} \tag{29}$$
$$= -\frac{3.257 \times 10^5 \text{ km}^{\cancel{3}\,2}/\text{sec}^2}{9828.5807 \cancel{\text{ km}}}$$
$$\mathcal{E} = -16.5690 \text{ km}^2/\text{sec}^2$$

FROM EQN. (7),

$$V_A = \sqrt{2\left(\frac{\mu_\oplus}{r_A} + \mathcal{E}\right)} \tag{30}$$
$$= \sqrt{2\left(\frac{3.257 \times 10^5 \text{ km}^3/\text{sec}^2}{13{,}070.1614 \cancel{\text{ km}}} - 15.569 \frac{\text{km}^2}{\text{sec}^2}\right)}$$
$$V_A = 4.0866 \text{ km/sec}$$

SIMILARLY,

$$V_P = \sqrt{2\left(\frac{\mu_\oplus}{r_P} + \mathcal{E}\right)} \tag{31}$$
$$= \sqrt{2\left(\frac{3.257 \times 10^5 \text{ km}^3/\text{sec}^2}{6587 \text{ km}} - 15.569 \frac{\text{km}^2}{\text{sec}^2}\right)}$$
$$V_P = 8.1089 \text{ km/sec}$$

THUS

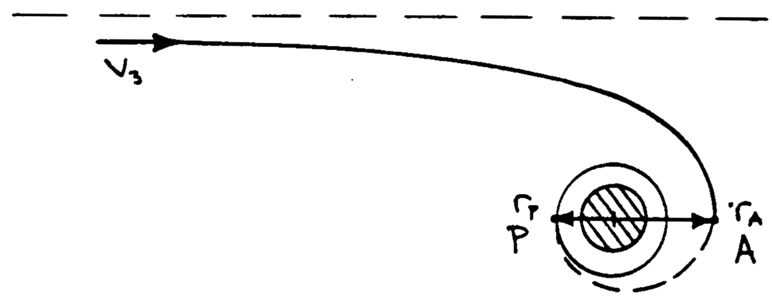
$$\underline{\underline{\Delta V_A = V_A - V_{cs_A} = (-)\,0.9053 \text{ km/sec}}} \quad \text{(RETRO)} \tag{32}$$

$$\underline{\underline{\Delta V_P = V_{cs_P} - V_P = (-)\,1.0771 \text{ km/sec}}} \quad \text{(RETRO)} \tag{33}$$

FOR THE 3-BURN MANEUVER TO CIRCULAR ORBIT OF 6587 km RADIUS, THE MER IS

$$\Delta V_{c_3} = \Delta V_{P_\oplus} + \Delta V_A + \Delta V_P \tag{34}$$
$$\boxed{\Delta V_{c_3} = 4.5527 \text{ km/sec}}\,.$$

(2) CIRCULAR ORBIT IS OBTAINED BY 2 BURNS IN THE FOLLOWING SEQUENCE:



$r_A = r_{P\oplus} = 13,070.1614$ km

$V_{P\oplus} = 7.5662$ km/sec

$\mathcal{E} = -15.569$ km²/sec²

FROM EQN. (30), $V_A = \sqrt{2\left(\frac{\mu_\oplus}{r_A} + \mathcal{E}\right)}$ (35)

$= 4.0866$ km/sec,

FROM EQN. (31), $V_P = \sqrt{2\left(\frac{\mu_\oplus}{r_P} + \mathcal{E}\right)}$ (36)

$= 8.1089$ km/sec,

THEN $\Delta V_A = V_A - V_{P\oplus} = (-)3.4756$ km/sec (37)

$\Delta V_P = V_{cs_P} - V_P = (-)1.0771$ km/sec (38)

FOR THE 2-BURN MANEUVER INTO A CIRCULAR ORBIT OF RADIUS 6587 km, MER IS

$$\Delta V_{c_2} = \Delta V_A + \Delta V_P \quad (39)$$

$$\boxed{\Delta V_{c_2} = (-)\,4.5527 \text{ km/sec}}$$

THE 3-BURN AND 2-BURN MANEUVERS SHOW THE SAME ENERGY REQUIREMENT; HOWEVER, THE

3-BURN ADVANTAGE IS THAT THE AVERAGE AMOUNT OF ENERGY PER BURN IS NOTICEABLY SMALLER THAN THAT OF THE 2-BURN MANEUVER, AS SHOWN BELOW.

- 3-BURN :

$$\Delta V_{ANG/BURN} = \frac{\Delta V_{C_3}}{3 \text{ BURNS}} = \frac{(-)\,4.5527 \text{ km/sec}}{3 \text{ BURNS}}$$
$$\underline{\underline{\Delta V_{ANG/BURN} = (-)\,1.5176 \text{ km/sec/BURN}}} \tag{40}$$

- 2-BURN :

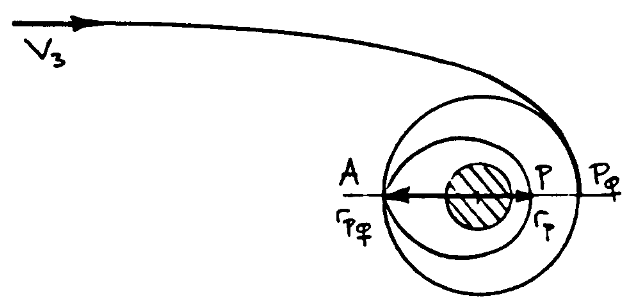
$$\Delta V_{ANG/BURN} = \frac{\Delta V_{C_2}}{2 \text{ BURNS}} = \frac{(-)\,4.5527 \text{ km/sec}}{2 \text{ BURNS}}$$
$$\underline{\underline{\Delta V_{ANG/BURN} = (-)\,2.2764 \text{ km/sec/BURN}}} \,. \tag{41}$$

$\Delta V_{ANG/BURN}$ FOR 3-BURN MANEUVER IS 33.33% LOWER THAN THAT FOR THE 2-BURN MANEUVER. A SMALLER BURN MEANS LESS OPPORTUNITY TO PERTURB THE VEHICLE IN FLIGHT FROM ITS INTENDED PATH. LOW ENERGIES ARE ESSENTIAL IF g-LOAD LIMITS ARE RESTRICTED TO LOW VALUES, SO THAT STRESSES DEVELOPED DURING BURN MANEUVERS DO NOT EXCEED THE VEHICLE'S STRUCTURAL DESIGN LIMIT.

THUS FAR, THE 3-BURN MANEUVER TO CIRCULARIZE THE ORBIT AT r_p OF 6587 km APPEARS TO SATISFY MER FOR ENERGY PER BURN.

FURTHER CONSIDERATION OF MER REQUIRES INVESTIGATION OF ESTABLISHING ELLIPTICAL ORBIT VIA BURNS SIMILAR TO THOSE OF THE PREVIOUS ORBITAL EVALUATION.

(3) ELLIPTICAL ORBIT IS OBTAINED IN 2 BURNS AS FOLLOWS :



$r_{p♂} = 13{,}070.1614$ km

$V_{p♂} = 7.5622$ km/sec

$r_p = 6587$ km/sec

FROM EQN. (27) , $V_{cs} = \sqrt{\frac{\mu_{♂}}{r_{p♂}}} = 4.9919 \ \frac{km}{sec}$.

AT $P_{♂}$: $\Delta V_{p♂} = V_{cs} - V_{p♂}$ FROM EQN. (28)

$$\Delta V_{p♂} = (-)\ 2.5703 \ \frac{km}{sec}$$

THEN USING A MINIMUM ENERGY TRANSFER TO OBTAIN THE MAPPING ALTITUDE OF 400 km AT PERIGEE ,

$$\mathcal{E} = -\frac{\mu_{♂}}{2a} \qquad \text{FROM EQN. (4)} \qquad (42)$$

$$= -\frac{3.257 \times 10^6 \ km^3/sec^2}{9828.5807 \ km}$$

$$\mathcal{E} = -16.569 \ \frac{km^2}{sec^2} .$$

FROM RESULT OF EQN. (35),

$$V_A = \sqrt{2\left(\frac{\mu_\oplus}{r_A} + \mathcal{E}\right)} = 4.0886 \text{ km/sec},$$

AND $\underline{\underline{\Delta V_A = V_A - V_{CS} = (-)\,0.9053 \text{ km/sec}}}$. (43)

FINALLY, THE ΔV_E REQUIRED TO ESTABLISH AN ELLIPTICAL ORBIT OF 6587 km AT PERIGEE IS,

$$\Delta V_E = \Delta V_{P_f} + \Delta V_A \tag{44}$$

$$\boxed{\Delta V_E = (-)\,3.4756 \text{ km/sec}}.$$

THIS 2-BURN ELLIPTICAL ORBIT MANEUVER IS SIMILAR TO THE 3-BURN CIRCULAR MANEUVER OF PART (1) WITHOUT THE FINAL BURN TO CIRCULARIZE AT 6587 km RADIUS. COMPARED TO THE 3-BURN CIRCULAR MANEUVER, THE 2-BURN ELLIPTICAL MANEUVER HAS A 23.65% ENERGY REDUCTION, AND HAS ONLY A SLIGHTLY HIGHER AVERAGE ENERGY PER BURN AS FOLLOWS:

$$\Delta V_{AVG/BURN} = \frac{\Delta V_E}{2 \text{ BURNS}} \tag{45}$$

$$= \frac{(-)\,3.4756 \text{ km/sec}}{2 \text{ BURNS}}$$

$$\underline{\underline{\Delta V_{AVG/BURN} = (-)\,1.7378 \text{ km/sec}}}.$$

SUMMARY: IF THE MISSION WERE JUDGED SOLELY ON THE BASIS OF MEETING MER, THEN THE 2-BURN ELLIPTICAL ORBIT MANEUVER WOULD BE THE CHOSEN OVER THE 3-BURN CIRCULAR ORBIT MANEUVER; HOWEVER, ONE OF NINEVAH'S PRIMARY OBJECTIVES IS TO MAP THE SURFACE OF VENUS. THE CIRCULAR ORBIT OF 6587 km RADIUS ENSURES COMPLETE COVERAGE OF SURFACE TOPOLOGY; WHEREAS THE ELLIPTICAL ORBIT ONLY MAPS WHEN WITHIN THE RADAR MAPPER'S RANGE (OF LESS THAN 550 km).

USING EQUATION (46), THE PERIOD OF THE TWO ORBITS IS CALCULATED.

$$\mathbb{P}_E = \frac{2\pi}{\sqrt{\mu_♀}} a^{3/2} \qquad \text{(BATE, ET AL: p. 33)} \qquad (46)$$

$$= \frac{2\pi}{\sqrt{3.257 \times 10^5 \text{ km}^3/\text{sec}^2}} (9828.5807 \text{ km})^{3/2}$$

$$\underline{\underline{\mathbb{P}_E = 10{,}727.7144 \text{ sec}_⊕ = 0.1242 \text{ DAYS}_⊕}}$$

$$\mathbb{P}_C = \frac{2\pi}{\sqrt{\mu_♀}} r_{cs}^{3/2} \qquad (47)$$

$$= \frac{2\pi}{\sqrt{3.257 \times 10^5 \text{ km}^3/\text{sec}^2}} (6587 \text{ km})^{3/2}$$

$$\underline{\underline{\mathbb{P}_C = 5{,}835.7591 \text{ sec} = 0.068 \text{ DAYS}_⊕}}$$

MATCHING THE PERIOD FOR THE ELLIPTICAL ORBIT WITH CYTHEREAN ROTATION TO GUARANTEE NOMINAL COVERAGE IS A COMPLEX PROBLEM; HOWEVER, THE CIRCULAR ORBIT WITH ITS CONSTANT

ALTITUDE IS ALWAYS IN THE MAPPER'S RANGE TO PROVIDE CONTINUOUS COVERAGE OF VENUS' SURFACE. AS IT TURNS OUT, NINEVAH IN CIRCULAR ORBIT AT THE PRESCRIBED ALTITUDE OF 400 km, MAKES ABOUT 14.5 ORBITS PER DAY, AS DETERMINED FROM EQUATION (48).

$$\#\text{ORBITS/DAY} = 1/P_c \quad (\text{INVERTING EQN. (47)}) \tag{48}$$

$$= \frac{1}{0.0681\ \text{day}}$$

$$\underline{\underline{\#\text{ORBITS/DAY} = 14.68\ /\text{DAY}}}$$

NINEVAH'S SURFACE MAPPING, AS THE CRAFT IS IN A CIRCULAR ORBIT, IS COMPLETE BY HALF THE ROTATION PERIOD (243 DAYS) OF VENUS, OR ABOUT 122 DAYS.

THE LOGICAL CHOICE FOR ORBITAL MANEUVERING IS, THEN, THE 3-BURN CIRCULAR ORBIT MANEUVER FOR THE REASONS DISCUSSED ABOVE.

	NITROGEN TETROXIDE	HYDRAZINE	UDMH
MOLECULAR WEIGHT	92.016	32.05	60.10
HEAT OF FORMATION (AT BOILING POINT)	413	1256	394
MELTING POINT	261.5 K.	274.5 K.	216 K.
BOILING POINT	294.3 K.	386.4 K.	336 K.
DENSITY (LB/FT**3)	94	98.4	76.2
	N_2O_4	N_2H_4	$(CH_3)_2N_2H_2$

I_{sp} = 310. sec

CATALYST--POTASSIUM CUPROCYANIDE $K_3Cu(CN)_4$

COMPATABLE STORAGE CONTAINER MATERIAL

STAINLESS STEEL
303
304
321
347

HYDRAZINE COMPOSITION

18% HYDRAZINE
82% NITRIC ACID HNO_3

OPTIMUM MIXTURE RATIO----1.2

STANDARD MIXTURE RATION--2.0

HYDRAZINE-UDMH/NITROGEN TETROXIDE I_{sp} = 250. sec

tanks

```
ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 21797.0

ENTER THE ISP VALUE OF THE PROPELLENT. 250.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA  V IS      15.02714060 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS  120325.93700000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS  128904.01600000 LBS.

ENTER THE  g  LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3



 AR =       75.00000000
 EXIT MACH NUMBER =       5.49001122
 PRESSURE RATIO =   6.08945207E-04
 CT =        1.83934772
 TEMPCT        1.82325983
 MAXIMUM THRUST =   14868.80270000 LBS.
 THRUST =   29737.60550000 LBS.
 GROSS WEIGHT =  128904.01600000 LBS.
 FUEL WEIGHT =  120325.93700000 LBS.
 TANK WEIGHT =    1978.08020000 LBS.
 TOTAL ENGINE(S) WEIGHT =      600.00000000 LBS.
 COMBUSTION CHAMBER PRESSURE =     150.00000000 PSI.
 MASS FLOW RATE =     118.95041700 LBM/SEC.
 DELTA T =    1011.56384000 SECONDS.
ENTER THE FUEL MIXTURE RATIO 2.0

ENTER THE NUMBER OF PAIRS OF PROPELLANT TANKS. 1.0
```

```
ENTER THE MAXIMUM LENGTH OF THE TANKS. 15.0



THE TOTAL PAIRS OF TANKS IS       1.00000000 .
***********************************

THE FUEL TANK DIMENSIONS ARE AS FOLLOWS:
TOTAL FUEL TANK VOLUME =     907.52453600 FT**3.
TANK LENGTH =        15.00000000 FT.
TANK RADIUS =         4.38842440 FT.
***********************************

THE OXID TANK DIMENSIONS ARE AS FOLLOWS:

TOTAL OXID TANK VOLUME =     426.68084700 FT**3.
TANK LENGTH =        15.00000000 FT.
TANK RADIUS =         3.00906110 FT.
***********************************


THE TOTAL FUEL BURNED =  120325.93700000 LBS.

C:\SCRATCH>
```

HYDRAZINE-UDMH/NITROGEN TETROXIDE I_{sp} = 275. sec

tanks

```
ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 21797.0

ENTER THE ISP VALUE OF THE PROPELLENT. 275.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA  V IS      11.74593450
THE WEIGHT OF THE FUEL FOR THE MISSION IS   84272.88280000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS   92115.18750000 LBS.

ENTER THE  g  LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3




 AR =       75.00000000
 EXIT MACH NUMBER =       5.49001122
 PRESSURE RATIO =   6.08945207E-04
 CT =       1.83934772
 TEMPCT       1.82325983
 MAXIMUM THRUST =   14868.80270000 LBS.
 THRUST =   29737.60550000 LBS.
 GROSS WEIGHT =   92115.18750000 LBS.
 FUEL WEIGHT =   84272.88280000 LBS.
 TANK WEIGHT =    1242.30359000 LBS.
 TOTAL ENGINE(S) WEIGHT =     600.00000000 LBS.
 COMBUSTION CHAMBER PRESSURE =     150.00000000 PSI.
 MASS FLOW RATE =     108.13674200 LBM/SEC.
 DELTA T =     779.31774900 SECONDS.
ENTER THE FUEL MIXTURE RATIO 2.0

ENTER THE NUMBER OF PAIRS OF PROPELLANT TANKS. 1.0

ENTER THE MAXIMUM LENGTH OF THE TANKS. 15.0
```

```
THE TOTAL PAIRS OF TANKS IS       1.00000000 .
***********************************

THE FUEL TANK DIMENSIONS ARE AS FOLLOWS:
TOTAL FUEL TANK VOLUME =      635.59637500 FT**3.
TANK LENGTH =       15.00000000 FT.
TANK RADIUS =        3.67257047 FT.
***********************************

THE OXID TANK DIMENSIONS ARE AS FOLLOWS:

TOTAL OXID TANK VOLUME =      298.82977300 FT**3.
TANK LENGTH =       15.00000000 FT.
TANK RADIUS =        2.51820660 FT.
***********************************

THE TOTAL FUEL BURNED =   84272.88280000 LBS.
```

ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 21797.0

ENTER THE ISP VALUE OF THE PROPELLENT. 310.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA V IS 8.89393234 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS 57611.43360000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS 64909.62500000 LBS.

ENTER THE g LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3

```
AR =        75.00000000
EXIT MACH NUMBER =         5.49001122
PRESSURE RATIO =    6.089452O7E-04
CT =        1.83934772
TEMPCT        1.82325983
MAXIMUM THRUST =    14868.80270000 LBS.
THRUST =    29737.60550000 LBS.
GROSS WEIGHT =    64909.62500000 LBS.
FUEL WEIGHT =    57611.43360000 LBS.
TANK WEIGHT =      698.19250500 LBS.
TOTAL ENGINE(S) WEIGHT =      600.00000000 LBS.
COMBUSTION CHAMBER PRESSURE =      150.00000000 PSI.
MASS FLOW RATE =        95.92775730 LBM/SEC.
DELTA T =      600.57104500 SECONDS.
```

ENTER THE FUEL MIXTURE RATIO 2.0

ENTER THE NUMBER OF PAIRS OF PROPELLANT TANKS. 1.0

ENTER THE MAXIMUM LENGTH OF THE TANKS. 15.0

```
THE TOTAL PAIRS OF TANKS IS       1.00000000 .
***********************************

THE FUEL TANK DIMENSIONS ARE AS FOLLOWS:
TOTAL FUEL TANK VOLUME =     434.51257300 FT**3.
TANK LENGTH =        15.00000000 FT.
TANK RADIUS =         3.03655124 FT.
***********************************

THE OXID TANK DIMENSIONS ARE AS FOLLOWS:

TOTAL OXID TANK VOLUME =     204.28723100 FT**3.
TANK LENGTH =        15.00000000 FT.
TANK RADIUS =         2.08209252 FT.
***********************************


THE TOTAL FUEL BURNED =   57611.43360000 LBS.

C:\SCRATCH>
```

LOX/H_2 I_{sp} = 350. sec

ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 21797.0

ENTER THE ISP VALUE OF THE PROPELLENT. 350.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA V IS 6.92828846 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS 41291.28910000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS 48256.41800000 LBS.

ENTER THE g LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3

AR = 75.00000000
EXIT MACH NUMBER = 5.49001122
PRESSURE RATIO = 6.08945207E-04
CT = 1.83934772
TEMPCT 1.82325983
MAXIMUM THRUST = 14868.80270000 LBS.
THRUST = 29737.60350000 LBS.
GROSS WEIGHT = 48256.41800000 LBS.
FUEL WEIGHT = 41291.28910000 LBS.
TANK WEIGHT = 365.12832600 LBS.
TOTAL ENGINE(S) WEIGHT = 600.00000000 LBS.
COMBUSTION CHAMBER PRESSURE = 150.00000000 PSI.
MASS FLOW RATE = 84.96458440 LBM/SEC.
DELTA T = 485.98236100 SECONDS.
ENTER THE FUEL MIXTURE RATIO 6.0

ENTER THE NUMBER OF PAIRS OF PROPELLANT TANKS. 1.0

ENTER THE MAXIMUM LENGTH OF THE TANKS. 15.0

```
THE TOTAL PAIRS OF TANKS IS       1.00000000 .
**********************************

THE FUEL TANK DIMENSIONS ARE AS FOLLOWS:
TOTAL FUEL TANK VOLUME =    5684.54883000 FT**3.
TANK LENGTH =       15.00000000 FT.
TANK RADIUS =       10.98316290 FT.
**********************************

THE OXID TANK DIMENSIONS ARE AS FOLLOWS:

TOTAL OXID TANK VOLUME =      69.88648990 FT**3.
TANK LENGTH =       15.00000000 FT.
TANK RADIUS =        1.21780026 FT.
**********************************


THE TOTAL FUEL BURNED =   41291.28910000 LBS.
```

maximum allowable fuel flow rate determination

$$\Delta V = U_e \ Ln\left(\frac{W_o}{W_b}\right)$$

where W_o = INITIAL Mass of Vehicle
W_b = Mass of vehicle at terminal velocITY

$$U_e = I_{sp} \ U_e$$

$\therefore$

$$\Delta V = I_{sp} \ g_c \ Ln\left(\frac{W_o}{W_b}\right)$$

$$m_o = m_b \ e^{\left(\frac{W_o}{W_b}\right)}$$

$$\Delta t = \frac{W_{Fuel}}{\dot{W}}$$

$$\frac{\Delta V}{\Delta t} = g_{Limit}$$

$$\Delta V = g_{LimiT} \ \Delta t$$

$$\Delta V = g_{Limit} \ \frac{W_{Fuel}}{\dot{W}}$$

$$\dot{W} = \frac{g_{LimiT} \ W_{Fuel}}{\Delta V}$$

$\therefore$ $\dot{W}$ is The total weight FLow rate required for The given ΔV at the g_{LimiT} acceleration

ORIGINAL PAGE IS
OF POOR QUALITY

ORIGINAL PAGE IS
OF POOR QUALITY

Fuel Weight determination

$$\Delta V = V_{eq} \ln\left(\frac{W_0}{W_b}\right)$$

$$\frac{W_0}{W_b} = e^{\left(\frac{\Delta V}{V_{eq}}\right)}$$

$$W_0 = W_{bus} + W_{fuel} + W_{tanks} + W_{engine}$$

$$W_b = W_{bus} + W_{tanks} + W_{engine} + (W_{fuel} - \dot{W}\Delta t)$$

Computation in the Program assume that $W_{fuel} - \dot{W}\Delta t = 0$

$\therefore$

$$\frac{W_0}{W_b} = \frac{W_{bus} + W_{tanks} + W_{engine} + W_{fuel}}{W_{bus} + W_{engine} + W_{tanks}}$$

$$\frac{W_0}{W_b} = \frac{W_{bus} + W_{tanks} + W_{engine}}{W_{bus} + W_{tanks} + W_{engine}} + \frac{W_{fuel}}{W_{bus} + W_{tanks} + W_{engine}}$$

$$\frac{W_0}{W_b} = 1 + \frac{W_{fuel}}{W_{bus} + W_{tanks} + W_{engine}}$$

$$\frac{W_0}{W_b} - 1 = \frac{W_{fuel}}{W_{bus} + W_{tanks} + W_{engine}}$$

$$W_{fuel} = \left(\frac{W_0}{W_b} - 1\right)\left(W_{bus} + W_{tanks} + W_{engine}\right)$$

$$W_{fuel} - W_{tanks}\left(\frac{W_0}{W_b} - 1\right) = \left(\frac{W_0}{W_b} - 1\right)\left(W_{bus} + W_{engine}\right) \quad (1)$$

ORIGINAL PAGE IS
OF POOR QUALITY

fuel Weight Determination (Continued)

assuming That the propulsion system (tanks and engines) Compose approximately 2 % of the total weight of the launch vehicle :

$$W_{tanks} + W_{engine} = .05\ W_0$$

$$W_{tanks} + W_{engine} = .05\ (W_{tanks} + W_{engine} + W_{fuel} + W_{Bus})$$

$$.95\ W_{tanks} - .05\ W_{fuel} = .05\ (W_{engine} + W_{Bus}) - W_{engine}$$

$$.95\ W_{tanks} - .05\ W_{fuel} = -.95\ W_{engine} + .05\ W_{Bus} \quad (2)$$

Solving eq (1) and eq (2) Simultaneously:

$$W_{fuel} - W_{tanks}\left(\frac{W_0}{W_b} - 1\right) = \left(\frac{W_0}{W_b} - 1\right)\left(W_{bus} + W_{engine}\right)$$

$$-.05\ W_{fuel} + .95\ W_{tanks} = -.95\ W_{engines} + .05\ W_{Bus}$$

$$-.05\ W_{tanks}\left(\frac{W_0}{W_b} - 1\right) + .95\ W_{tanks} = -.95\ W_{engine} + .05\ W_{Bus} + .05\left(\frac{W_0}{W_b} - 1\right)\left(W_{Bus} + W_{engine}\right)$$

$$W_{tanks}\left(-.05\left(\frac{W_0}{W_b} - 1\right) + .95\right) = -.95\ W_{engine} + .05\ BUS + .05\left(\frac{W_0}{W_b} - 1\right)\left(W_{bus} + W_{engine}\right).$$

$$W_{tanks} = \frac{-.95\ W_{engine} + .05\ W_{BUS} + .05\left(\frac{W_0}{W_b} - 1\right)\left(W_{bus} + W_{engine}\right)}{\left(-.05\left(\frac{W_0}{W_b} - 1\right) + .95\right)}$$

ORIGINAL PAGE IS
OF POOR QUALITY

Fuel Weight Determination (Continued)

$$W_{fuel} = \left(\frac{W_e}{W_b} - 1 \right)\left(W_{bus} + W_{tanks} + W_{engine} \right)$$

LOX/H_2 I_{sp} = 455. sec

```
ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 21797.0

ENTER THE ISP VALUE OF THE PROPELLENT. 455.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA  V IS        4.43236303 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS   22597.37300000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS   29180.99410000 LBS.

ENTER THE  g  LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3


 AR =       75.00000000
 EXIT MACH NUMBER =       5.49001122
 PRESSURE RATIO =   6.08945207E-04
 CT =       1.83934772
 TEMPCT       1.82325983
 MAXIMUM THRUST =   14868.80270000 LBS.
 THRUST =   29737.60350000 LBS.
 GROSS WEIGHT =   29180.99410000 LBS.
 FUEL WEIGHT =   22597.37300000 LBS.
 TANK WEIGHT =     -16.38016700 LBS.
 TOTAL ENGINE(S) WEIGHT =     600.00000000 LBS.
 COMBUSTION CHAMBER PRESSURE =     150.00000000 PSI.
 MASS FLOW RATE =     65.35736850 LBM/SEC.
 DELTA T =    345.75097700 SECONDS.
ENTER THE FUEL MIXTURE RATIO 6.0

ENTER THE NUMBER OF PAIRS OF PROPELLANT TANKS. 1.0

ENTER THE MAXIMUM LENGTH OF THE TANKS. 15.0
```

```
THE TOTAL PAIRS OF TANKS IS       1.00000000 .
***********************************

THE FUEL TANK DIMENSIONS ARE AS FOLLOWS:
TOTAL FUEL TANK VOLUME =     3110.98633000 FT**3.
TANK LENGTH =        15.00000000 FT.
TANK RADIUS =         8.12509537 FT.
***********************************

THE OXID TANK DIMENSIONS ARE AS FOLLOWS:

TOTAL OXID TANK VOLUME =       38.24916460 FT**3.
TANK LENGTH =        15.00000000 FT.
TANK RADIUS =         0.90092868 FT.
***********************************


THE TOTAL FUEL BURNED =    22597.37300000 LBS.
```

BURN TIME   DEPARTURE FROM EARTH

```
ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 8432.75

ENTER THE ISP VALUE OF THE PROPELLENT. 310.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA  V IS      2.32906818 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS    8364.01660000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS   14657.15920000 LBS.

ENTER THE  g  LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3


*****************************************
THE NUMBER OF ENGINES AT FULL THRUST IS
TOO LARGE.  REDUCING MASS FLOW RATE TO
BRING WITHIN  g  LIMIT.
*****************************************


AR =       75.00000000
EXIT MACH NUMBER =        5.49001122
PRESSURE RATIO =   6.08945207E-04
CT =       1.83934772
TEMPCT       1.82325983
MAXIMUM THRUST =   14868.80270000 LBS.
THRUST =   57492.70310000 LBS.
GROSS WEIGHT =   14657.15920000 LBS.
FUEL WEIGHT =    8364.01660000 LBS.
TANK WEIGHT =    -306.85681200 LBS.
TOTAL ENGINE(S) WEIGHT =     600.00000000 LBS.
COMBUSTION CHAMBER PRESSURE =     145.00000000 PSI.
MASS FLOW RATE =        92.73016360 LBM/SEC.
DELTA T =      90.19736480 SECONDS.
```

BURN TIME   MID-COURSE CORRECTION

```
ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 526.738

ENTER THE ISP VALUE OF THE PROPELLENT. 310.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA  V IS       1.05423021 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS     332.38928200 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS    6461.62158000 LBS.

ENTER THE  g  LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 050.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.8671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3



******************************************
THE NUMBER OF ENGINES AT FULL THRUST IS
TOO LARGE.  REDUCING MASS FLOW RATE TO
BRING WITHIN  g  LIMIT.
******************************************



AR =       75.00000000
EXIT MACH NUMBER =        5.49001122
PRESSURE RATIO =   6.08945207E-04
CT =       1.83934772
TEMPCT       1.82325983
MAXIMUM THRUST =   14868.80270000 LBS.
THRUST =   37667.63280000 LBS.
GROSS WEIGHT =    6461.62158000 LBS.
FUEL WEIGHT =     332.38928200 LBS.
TANK WEIGHT =    -470.76757800 LBS.
TOTAL ENGINE(S) WEIGHT =     600.00000000 LBS.
COMBUSTION CHAMBER PRESSURE =      95.00000000 PSI.
MASS FLOW RATE =      60.75424580 LBM/SEC.
DELTA T =       5.47104597 SECONDS.
```

BURN TIME   ORBITAL INSERTION-VENUS

```
C:\AYER>6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 6334.31

ENTER THE ISP VALUE OF THE PROPELLENT. 310.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA  V IS       1.88717031 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS    5531.81104000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE,IS    11767.15430000 LBS.

ENTER THE  g  LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3


*****************************************
THE NUMBER OF ENGINES AT FULL THRUST IS
TOO LARGE.  REDUCING MASS FLOW RATE TO
BRING WITHIN  g  LIMIT.
*****************************************


AR =        75.00000000
EXIT MACH NUMBER =        5.49001122
PRESSURE RATIO =   6.08945207E-04
CT =        1.83934772
TEMPCT        1.82325983
MAXIMUM THRUST =   14868.80270000 LBS.
THRUST =   51545.18360000 LBS.
GROSS WEIGHT =    11767.15430000 LBS.
FUEL WEIGHT =    5531.81104000 LBS.
TANK WEIGHT =    -364.65692100 LBS.
TOTAL ENGINE(S) WEIGHT =      600.00000000 LBS.
COMBUSTION CHAMBER PRESSURE =      130.00000000 PSI.
MASS FLOW RATE =      83.13739010 LBM/SEC.
DELTA T =      66.53818510 SECONDS.
```

BURN TIME ELLIPTICAL TRANSFER-LOWER CYTHEREAN ORBIT

```
ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 3533.79

ENTER THE ISP VALUE OF THE PROPELLENT. 310.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA  V IS       1.42517996 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS    2625.92822000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS    8801.96777000 LBS.

ENTER THE  g  LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3


******************************************
THE NUMBER OF ENGINES AT FULL THRUST IS
TOO LARGE.  REDUCING MASS FLOW RATE TO
BRING WITHIN  g  LIMIT.
******************************************


AR =       75.00000000
EXIT MACH NUMBER =        5.49001122
PRESSURE RATIO =    6.08945207E-04
CT =      1.83934772
TEMPCT        1.82325983
MAXIMUM THRUST =    14868.80270000 LBS.
THRUST =    43615.15620000 LBS.
GROSS WEIGHT =     8801.96777000 LBS.
FUEL WEIGHT =     2625.92822000 LBS.
TANK WEIGHT =     -423.96066300 LBS.
TOTAL ENGINE(S) WEIGHT =       600.00000000 LBS.
COMBUSTION CHAMBER PRESSURE =      110.00000000 PSI.
MASS FLOW RATE =        70.34702300 LBM/SEC.
DELTA T =       37.32820510 SECONDS.
```

BURN TIME CIRCULARIZATION-LOWER CYTHEREAN ORBIT

ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) 6000.0

ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) 300.0

ENTER THE NUMBER OF ENGINES TO BE USED. 2.0

ENTER THE DELTA V OF THE MISSION. (FT/SEC) 2970.14

ENTER THE ISP VALUE OF THE PROPELLENT. 310.0

THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA V IS 1.34687424 .
THE WEIGHT OF THE FUEL FOR THE MISSION IS 2138.86084000 LBS.
THE TOTAL WEIGHT OF THE VEHICLE IS 8304.95996000 LBS.

ENTER THE g LIMIT OF THE STRUCTURE. 3.0

ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS 15000.0

ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) 150.0

ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) 54.3671

ENTER THE INITIAL AREA RATIO. 75.0

ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. 1.3

****
THE NUMBER OF ENGINES AT FULL THRUST IS
TOO LARGE. REDUCING MASS FLOW RATE TO
BRING WITHIN g LIMIT.
****

AR = 75.00000000
EXIT MACH NUMBER = 5.49001122
PRESSURE RATIO = 6.08945207E-04
CT = 1.83934772
TEMPCT 1.82325983
MAXIMUM THRUST = 14868.80270000 LBS.
THRUST = 41632.64840000 LBS.
GROSS WEIGHT = 8304.95996000 LBS.
FUEL WEIGHT = 2138.86084000 LBS.
TANK WEIGHT = -433.90081800 LBS.
TOTAL ENGINE(S) WEIGHT = 600.00000000 LBS.
COMBUSTION CHAMBER PRESSURE = 105.00000000 PSI.
MASS FLOW RATE = 67.14942930 LBM/SEC.
DELTA T = 31.85225680 SECONDS.

```
C:\FORT>C      THIS PROGRAM WAS WRITTEN FOR SENIOR DESIGN 448 AND WAS USED TO
C      DETERMINE THE AMOUNT OF FUEL NEEDED FOR THE MISSION PROFILE.
C      WRITTEN BY WAYNE AYER 17-FEBRUARY 1988.
C*****************************************************************
C                            DEFINITIONS
C
C      AR  AREA RATIO OF THE NOZZLE
C      CSTAR  CHARACTERISTIC VELOCITY
C      CT  THRUST COEFFICIENT
C      DELTAT  BURN TIME TO ACHIEVE GIVEN DELTA V
C      DELTAV  DELTA V REQUIRED FOR NAVIGATIONAL MANUEVER
C      FISP  FUEL ISP
C      GAMMA  GAMMA OF THE FUEL
C      GC  ACCELERATION DUE TO GRAVITY
C      GLIMIT  LIMIT G THE SPACE VEHICLE MAY SEE
C      ME  MACH NUMBER AT THE EXIT OF THE NOZZLE
C      PE  PRESSURE AT THE EXIT OF THE NOZZLE
C      PR  PRESSURE RATIO OF NOZZLE:PT2/PE
C      PT2  PRESSURE OF COMBUSTION CHAMBER
C      R   MIXTURE RATIO OF THE FUEL AND OXIDIZER
C      TEMPAR  TEMPORARY AREA RATIO:USED IN ITERATION OF ME
C      THROAT  THROAT AREA OF THE NOZZLE
C      THRUST  THRUST OF THE ENGINE(S)
C      THRUSTREQUIRED  THRUST REQUIRED TO ACHIEVE DELTA V.
C      UEQ EQUIVALENT EXHAUST VELOCITY
C      WBUS  WEIGHT OF SPACE VEHICLE WITHOUT PROPULSION SYSTEM.
C      WENGINE WEIGHT OF ROCKET ENGINE
C      WFUEL   WEIGHT OF FUEL
C      WTANKS  WEIGHT OF TANKS
C      WTOTAL  TOTAL WEIGHT OF VEHICLE AT BEGINNING OF MISSION PROFILE
C*********************************************************************
C
C
C
C
        REAL ME,MASSFLOW,MASSFLOWNEEDED
        OPEN(UNIT=66,FILE='A:TANKS.DAT',STATUS='OLD')
        REWIND 66
        GC=32.17438
        PI = 3.14159265
        WRITE(6,*)'ENTER THE WEIGHT OF THE SPACE VEHICLE.(LBS) '
        READ(5,*)WBUS
C       WRITE(6,*)'ENTER THE WEIGHT OF THE FUEL.(LBS) '
C       READ(5,*)WFUEL
        WRITE(6,*)'ENTER THE WEIGHT OF A PROPULSION SYSTEM.(LBS) '
        READ(5,*)WENGINE
        WRITE(6,*)'ENTER THE NUMBER OF ENGINES TO BE USED. '
        READ(5,*)ENGINENBR
        WENGINE = WENGINE*ENGINENBR

C*****************************************************************
C
C     THE PROPULSION SYSTEM INCLUDING THE FUEL TANKS IS ASSUMED TO BE
C     APPROXIMATELY 2% OF THE TOTAL WEIGHT OF THE VEHICLE AT MISSION
C     INTITIATION. THEREFORE THE TANK WEIGHT IS CALCULATED AS FOLLOWS:
C
C*******************************************************************
        WRITE(6,*)'ENTER THE DELTA V OF THE MISSION. (FT/SEC) '
        READ(5,*)DELTAV
        WRITE (6,*) 'ENTER THE ISP VALUE OF THE PROPELLENT. '
        READ (5,*)FISP
        UEQ = FISP*GC
C       WO = .4999
C       WB = .5001
C       DELTAUPPER = DELTAV + 100.0
```


ORIGINAL PAGE IS
OF POOR QUALITY

```
C         DELTALOWER = DELTAV - 100.0
C         DO 5 I = 1,1000000
C         WO = WO + .0001
C         WB = WB - .0001
C         TESTDELTAV = UEQ*EXP(WO/WB)
C         WR = WO/WB
C         WRITE(6,*)'WR ',WR
C         IF (TESTDELTAV .GT. DELTAV) GOTO 6
C   5     CONTINUE
          WR = EXP(DELTAV/UEQ)
C   6     WR = WO/WB
          WTANKS = (-.98*WENGINE+.02*WBUS+.02*(WR-1.0)*(WBUS+WENGINE))
     &              /(-.02*(WR-1.0)+.98)
          WFUEL = (WR-1.0)*(WBUS + WTANKS + WENGINE)
C         WTANKS = (.80*(WBUS+WFUEL+WENGINE)-WENGINE)/.20
          WTOTAL = WBUS+WFUEL+WTANKS+WENGINE
         WRITE(6,*)'THE INVERSE WEIGHT RATIO REQUIRED FOR THE GIVEN DELTA
     &V IS ',WR, ' .'
          WRITE(6,*)'THE WEIGHT OF THE FUEL FOR THE MISSION IS ',WFUEL,
     & ' LBS.'
          WRITE(6,*)'THE TOTAL WEIGHT OF THE VEHICLE IS ',WTOTAL,' LBS.'
          WRITE(6,*)' '
          WRITE(6,*)'ENTER THE  g  LIMIT OF THE STRUCTURE. '
          READ(5,*)GLIMIT
C         DELTAT = DELTAV/GLIMIT
          WRITE(6,*)'ENTER THE THRUST RANGE FOR INITIAL CALCULATIONS '
          READ(5,*) THRUST
          WRITE(6,*)'ENTER THE CHAMBER PRESSURE OF THE ENGINE. (PSI) '
          READ(5,*)PT2
          WRITE(6,*)'ENTER THE THROAT AREA OF THE NOZZLE. (IN**2) '
          READ(5,*) THROAT
          CT = THRUST/(PT2*THROAT)
c         WRITE(6,*) 'ENTER THE TOLERANCE OF CT (+ OR -)  .'
c         READ(5,*)CONVERGENCE
C         IF(CT .GT. .6 .AND. CT .LT. 2.5) GO TO 100
C         WRITE (6,*)'CT = ',CT
C         WRITE (6,*)'THE RESULTING CT IS NOT WITHIN THE ANALYTICAL'
C         WRITE (6,*)'VALUE OF CT.  DECREASE THE VALUE OF PT2 OR THE '
C         WRITE (6,*)'THROAT AREA TO BRING THE VALUE WITHIN TOLERANCE.'
C         GOTO 110
C         DUMMY = 1
          WRITE(6,*)'ENTER THE INITIAL AREA RATIO. '
          READ(5,*)AR
          WRITE(6,*)'ENTER THE VALUE OF GAMMA FOR THE PROPELLANT. '
          READ(5,*)GAMMA
C ************************************************************
C
C    THE FOLLOWING PORTION OF THE PROGRAM ITERATES TO FIND THE CORRECT
C    VALUE OF THE AREA RATIO AND HENCE EXIT MACH NUMBER AND PRESSURE
C    RATIO.  THE PROGRAM ASSUMES A VALUE FOR THE AREA RATIO AND THEN
C    CALCULATES THE EXIT MACH NUMBER AND PRESSURE RATIO.  FROM THIS
C    THE PROGRAM CALCULATES CT AND COMPARES THE CALCULATED VALUE WITH
C    THAT OBTAINED FROM THE G LIMITATION ABOVE.  THE ITERATION CONTINUES
C    UNTIL THE AREA RATIO USED RESULTS IN A CT WITHIN 5% OF THE ABOVE
C    VALUE.
C
C**************************************************************
          TWOVERGAMMA = 2.0/(GAMMA + 1.0)
          GAMMASQRD = GAMMA ** 2.0
          GAMINONE= GAMMA - 1.0
          GAMPLONE = GAMMA + 1.0
          TWOGSQR= (2.0*GAMMA**2.0)/GAMINONE
          RATIO = GAMPLONE/GAMINONE
          RATIO1 = GAMINONE/GAMMA
          ME=5.0
```

ORIGINAL PAGE IS
OF POOR QUALITY

ORIGINAL PAGE IS
OF POOR QUALITY

```
      MASSFUEL = TOTFUELMASS ... ... ...
      MASSOXID = TOTFUELMASS*(1./(R+1.0))
      VOLFUEL  = MASSFUEL/RHOFUEL
      VOLXID   = MASSOXID/RHOXID
C
C
C    FUEL AND OXIDIZER TANKS ARE ASSUMMED TO BE CYLINDRICAL.
C
C
      WRITE(6,*) 'ENTER THE NUMBER OF PAIRS OF PROPELLANT TANKS. '
      READ(5,*) TANKNUMBER
      WRITE(6,*) 'ENTER THE MAXIMUM LENGTH OF THE TANKS. '
      READ(5,*) TANKLENGTH
C     WRITE(6,*) 'ENTER THE MAXIMUM DIAMETER OF THE TANKS. '
C     READ(5,*) TANKDIAMETER
C
C
C
      VOLXIDN = VOLXID/TANKNUMBER
      VOLFUELN = VOLFUEL/TANKNUMBER
      OXIDRADIUS = (VOLXIDN/TANKLENGTH/PI)**.5
      FUELRADIUS = (VOLFUELN/TANKLENGTH/PI)**.5
      WRITE(6,*) ' '
      WRITE(6,*) ' '
      WRITE(6,*) ' '
      WRITE(6,*) ' '
      WRITE(6,*) 'THE TOTAL PAIRS OF TANKS IS ',TANKNUMBER,' .'
      WRITE(6,*) '************************************'
      WRITE(6,*) ' '
      WRITE(6,*) 'THE FUEL TANK DIMENSIONS ARE AS FOLLOWS: '
      WRITE(6,*) 'TOTAL FUEL TANK VOLUME = ',VOLFUELN,' FT**3.'
      WRITE(6,*) 'TANK LENGTH = ',TANKLENGTH,' FT.'
      WRITE(6,*) 'TANK RADIUS = ',FUELRADIUS,' FT.'
      WRITE(6,*) '************************************'
      WRITE(6,*) ' '
      WRITE(6,*) 'THE OXID TANK DIMENSIONS ARE AS FOLLOWS: '
      WRITE(6,*) ' '
      WRITE(6,*) 'TOTAL OXID TANK VOLUME = ',VOLXIDN,' FT**3.'
      WRITE(6,*) 'TANK LENGTH = ',TANKLENGTH,' FT.'
      WRITE(6,*) 'TANK RADIUS = ',OXIDRADIUS,' FT.'
      WRITE(6,*) '************************************'
      WRITE(6,*) ' '
      WRITE(6,*) ' '
      WRITE(6,*)'THE TOTAL FUEL BURNED = ',TOTFUELMASS,' LBS.'
C     EXCESSFUEL = WFUEL - TOTFUELMASS
C     WRITE(6,*)'RESULTING IN ', EXCESSFUEL,' LBS OF EXCESS FUEL.'
      GOTO 10010
10000 WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' THE FUNCTION DID NOT CONVERGE. CHECK YOUR DATA.'
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
      WRITE (6,*) ' '
10010 CLOSE(UNIT = 66)
      END
```

```
C     OF THE AVAILABLE ENGINES.  THE ASSUMPTION IS USED IN THE FINAL
C     CONFIGURATION CHOSEN AND NOT DURING THE OPTIMIZATION AS FOLLOWS
C
C
          K = 0
          PTEST = PT2
          LENGINENBR = INT(ENGINENBR)
          DO 1005 I = 1,LENGINENBR
          ENGINES = REAL(LENGINENBR)
          DO 1000 J = 1,10000
          MASSFLOW = PTEST*THROAT*GC/CSTAR*ENGINES
C         WRITE(6,*) 'MASSFLOW ',MASSFLOW
          IF(MASSFLOW .LT. MASSFLOWNEEDED) GOTO 1010
          PTEST = PTEST - 5.0
          K = K + 1
          IF (K .GT. 1) GOTO 1000
          WRITE(6,*)' '
          WRITE(6,*)' '
          WRITE(6,*)' '
          WRITE(6,*)'*****************************************'
          WRITE(6,*)'THE NUMBER OF ENGINES AT FULL THRUST IS'
          WRITE(6,*)'TOO LARGE.  REDUCING MASS FLOW RATE TO'
          WRITE(6,*)'BRING WITHIN  g  LIMIT.'
          WRITE(6,*)'*****************************************'
          WRITE(6,*)' '
          WRITE(6,*)' '
          WRITE(6,*)' '
          DO 800 L = 1,100000
          R = R * 1.0
  800     CONTINUE
 1000     CONTINUE
 1005     CONTINUE
C         WRITE(6,*) 'ENGINES ',ENGINES
 1010     DELTAT = WFUEL/MASSFLOW
          THRUST = MASSFLOW/GC*CSTAR*TEMPCT
          THRUSTMAXIMUM = PT2*THROAT*TEMPCT
          WRITE (6,*) ' '
          WRITE (6,*) ' '
          WRITE (6,*) ' '
          WRITE (6,*) ' '
          WRITE (6,*) ' '
          WRITE (6,*) ' '
          WRITE (6,*) ' AR = ',AR
          WRITE (6,*) ' EXIT MACH NUMBER = ',ME
          WRITE (6,*) ' PRESSURE RATIO = ',PR
          WRITE (6,*) ' CT = ',CT
          WRITE (6,*) ' TEMPCT ',TEMPCT
          WRITE (6,*) ' MAXIMUM THRUST = ',THRUSTMAXIMUM,' LBS.'
          WRITE (6,*) ' THRUST = ',THRUST,' LBS.'
          WRITE (6,*) ' GROSS WEIGHT = ',WTOTAL,' LBS.'
          WRITE (6,*) ' FUEL WEIGHT = ',WFUEL,' LBS.'
          WRITE (6,*) ' TANK WEIGHT = ',WTANKS,' LBS.'
          WRITE (6,*) ' TOTAL ENGINE(S) WEIGHT = ',WENGINE,' LBS.'
          WRITE (6,*) ' COMBUSTION CHAMBER PRESSURE = ',PTEST, ' PSI.'
          WRITE (6,*) ' MASS FLOW RATE = ',MASSFLOW,' LBM/SEC.'
          WRITE (6,*) ' DELTA T = ',DELTAT,' SECONDS.'
          TOTFUELMASS = MASSFLOW*DELTAT
          WRITE(6,*) 'ENTER THE FUEL MIXTURE RATIO '
          READ(5,*) R
C     THE MIXTURE RATIO OF UDMH TO HYDRAZINE IS 1 AND THE MIXTURE
C     RATIO OF THIS PROPELLANT COMBINATION AND THE OXIDIZER IS VARIABLE
C     WITH 1.2 BEING THE OPTIMUM MIXTURE RATIO.
          RHOFUEL = 88.391
          RHOXID = 94.0
          RHOMIXTURE = (RHOXID*RHOFUEL*(1.0+R)/(R*RHOFUEL+RHOXID))
          VOLUMEPROPTOTAL = TOTFUELMASS/RHOMIXTURE
```


ORIGINAL PAGE IS
OF POOR QUALITY

External Tank Structure

There are two tank, but they are identical so the calculation were done just for one

Volume of fuel tank $- 217.25\ ft^3$

$$\text{Volume} = \pi (\text{radius})^2 \text{ length}$$

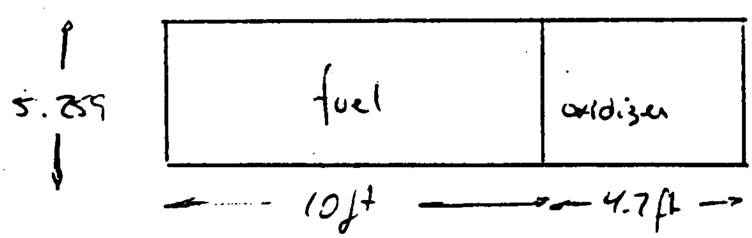
$$r_f = \sqrt{\frac{\text{Volume}}{\pi \text{ length}}} = \sqrt{\frac{217.25\ ft^3}{(3.14)(10\ ft)}}$$

$$d_f = 5.259\ ft$$

Volume of oxidizer per tank $= 102.14\ ft^3$

$$\text{Volume} = \pi r l$$

$$l = \frac{\text{Volume}}{\pi r^2} = \frac{102.14\ ft^3}{(3.14)(2.629\ ft} = 4.7\ ft$$



total weight of propellent is 64,909.62 lb at departure from earth orbit. At 3 g's acceleration

produces a force of 194,727 lb_f. For a cross-section area of the propellant tank of 3126.7 in^2, AISI 304 Stainless Steel was choosen. ($\sigma = 97{,}000\ lb/in^2$)

$$\sigma = F/A$$

$$A = F/\sigma = \frac{194{,}727\ lb_f}{97{,}000\ lb_f/in} = 2.00\ in^2$$

+10% safty gives $A = 2.2\ in^2$

internal cross section + A = 3128.9 in^2, this gives an external tank radius of 2.63 ft.

Now for an internal pressure of 75 lb/in^2

$$\sigma_L = \frac{D \Delta P}{2t}$$ where t is the thickness of the tank wall

$$t = \frac{D \Delta P}{2 \sigma_L} = \frac{(63.12\ in)(75\ lb/in^2)}{2(97{,}000\ lb/in^2)} = .0244\ in$$

This is about 10% of the thickness needed during axial loading, so the tank thickness due to tank pressurization will

ignored. For the internal pressure of 75 psi

$$\sigma_c = \frac{D\Delta P}{4t} \rightarrow t = \frac{D\Delta P}{4\sigma_c}$$

$$t = \frac{(63.12)(75\,psi)}{(2)(97{,}000\,lb)} = .0244\,in$$

thickness of the longitudinal tank skin

this is less than 1/10 acceleration (in the worst case) so it will be ignored.

Hemispherical end caps with the equation of

$$t = \frac{D\Delta P}{2\sigma_L}$$

During a 5g burn the end caps will a force of $184{,}727\,lb / 3128.9\,in^2 = 62.23\,lb/in$ acting on them.

$$\Delta P = 62.3 + 75 = 137.23\,lb/in^2$$

$$\sigma = \frac{D\Delta P}{2t} = \frac{(62.3)(137.23)}{2(2in)} = 1082.7\,lb/in^2$$

this is well within the stress limits of

**ORIGINAL PAGE IS
OF POOR QUALITY**

the metal choosen.

Hemispherical tank ends with an internal pressure of 75 psi:

$$\sigma_c = \frac{D \Delta P}{4t} \rightarrow t = \frac{D \Delta P}{4 \sigma_L}$$

$$t = \frac{(63.12\text{ in})(75\ \text{lb/in}^2)}{4(97{,}000\ \text{lb/in}^2)} = .0122\text{ in}$$

As before, this is less than 1/10 of the force that would be experienced during a 3g acceleration, so it will be ignored.

During the 3g burn the hemispherical end caps will have a force acting on them equal to:

$$F = 194{,}727\ \text{lb} / 3128.5\ \text{in}^2 = 62.23\ \text{lb/in}$$

$$\Delta F = \frac{D \Delta P}{4t} = \frac{(63.12)(75\text{ psi})}{4(2\text{ in})} = 591.75\ \text{lb}$$

this is also well within the stress limits of the metal choosen (AISI 304 Stainless Steel)

ORIGINAL PAGE IS
OF POOR QUALITY

Internal Tank

This tank is sandwitch between the two interment sections, provide fuel for course corrections and fuel as back-up for any type of emergency manuver.

Weight of propellent = 1988.8 lb_m

Weight of fuel = 1325.88 lb_m

Weight of oxidizer = 662.9 lb_m

Volume of fuel = 10 ft^3

Volume of oxidizer = 5 ft^3

$$l = \frac{Volume}{\pi r^2} = \frac{15\ ft^3}{\pi (4.635)^2} = .222\ ft$$

At 3g's acceleration gives a force of 5966.4 lb_f

Also using AISI 304 Stainless Steel.

$$A = F/\sigma = \frac{5966.4\ lb}{97{,}000\ lb/in^2} = .0615\ in^2$$

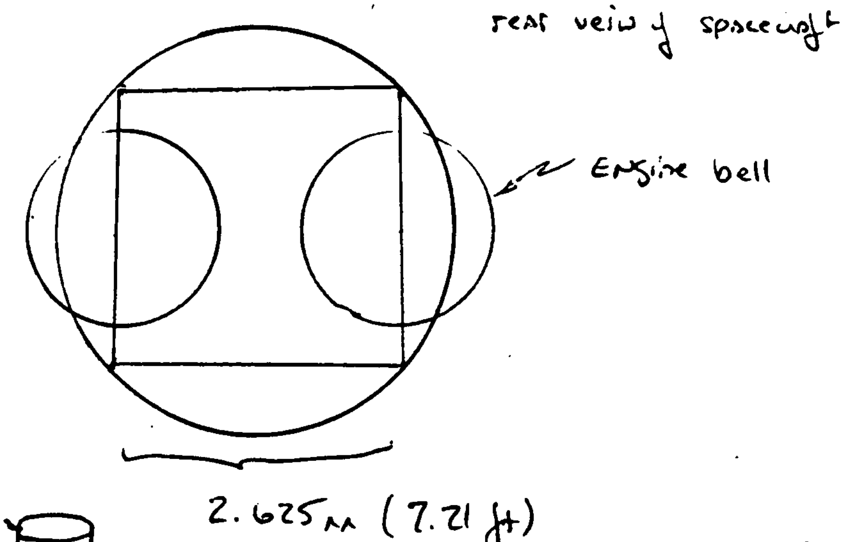
Cross-section of the tank + tank thickness give:

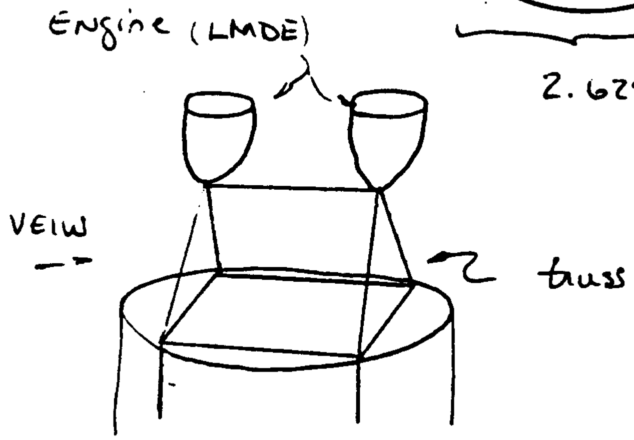
$A = 9718.7 \, in^2 + .0615 \, in^2$

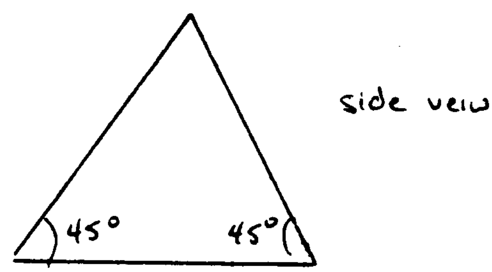
This gives an outside tank diameter of 9.271 ft. As with the external tanks, the stress from the fuel pressurization was 10% that of the force from the axial load and therefore was not taken into account. It is assumed that since tanks are stressed for an emergency burn of 3g's, they should be able to handle the ~75 psi pressurization with no problem, during routine phases of the mission.

ORIGINAL PAGE IS
OF POOR QUALITY

Engine Truss Analysis

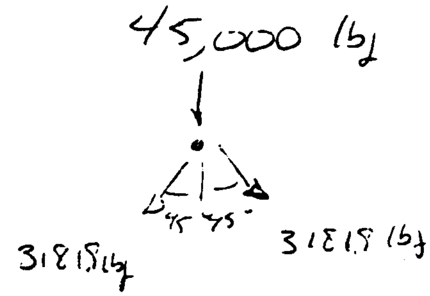


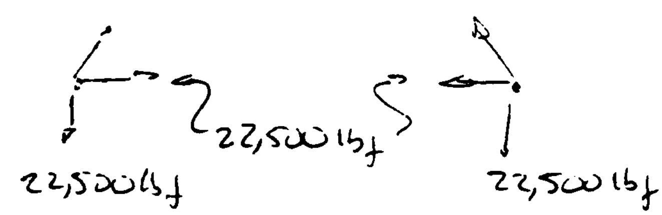




Since the truss is symmetric about the middle of the engine it is assumed that each side absorbs/transmits equal amounts of force to the truss structure

At 3g's the force excerted on the truss members





for cast Aluminum 2024-TA $\sigma = 48{,}000$ lb

$$A = F/\sigma = \frac{3181.9 \text{ lb}}{48{,}000 \text{ lb/in}^2} = .6628 \text{ in}^2$$

This gives a radius for a circular strut of .459 in with a factor of 10% safety this gives a diameter of the pyramidal truss structure of about one inch.

For the truss structure that joins the engine truss to the frame of the spacecraft:

ORIGINAL PAGE IS
OF POOR QUALITY

$$A = F/\sigma = 22{,}500 \text{ lb} / 48{,}000 \text{ lb/in}^2$$

$$A = .4687 \text{ in}^2$$

giving a radius of .386 inches for the spacecraft truss system.